EXPERIMENTAL INVESTIGATION OF A 2 1/8"-DIAMETER CONSTANT-AREA AREOTHERMOPRESSOR WITH SUPERSONIC INLET

Robert T. MacKay

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EXPERIMENTAL INVESTIGATION OF A 2 1/8"-DIAMETER CONSTANT-AREA AEROTHERMOPRESSOR WITH SUPERSONIC INLET

by

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Abstract

EXPERIMENTAL INVESTIGATION OF A 2 1/8"-DIAMETER CONSTANT-AREA AEROTHERMOPRESSOR WITH SUPERSONIC INLET

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Robert Torrey MacKay

Submitted to the Department of Mechanical Engineering on May 27, 1955 in partial fulfillment of the requirements for the degree of Master of Science in Mechanical Engineering.

It is known from theoretical studies that good performance of the Aerothermopressor, currently under study at the Massachusetts Institute of Technology under the direction of Professor A. H. Shapiro and under the sponsorship of the Office of Naval Research, is dependent upon the ability to produce extremely small droplets upon injecting cooling water into a hot, high-velocity gas stream.

The Aerothermopressor, a circular duct fitted with a water injection nozzle, produces a rise in the stagnation pressure of such a gas stream by the mechanism of evaporative cooling, and has as its main objective the improvement of gas turbine plant performance. To accomplish this objective it must necessarily rely upon a high evaporation rate.

By taking in gas-turbine exhaust gases, raising their stagnation pressure, and exhausting to atmospheric pressure, the Aerothermopressor will in effect produce a vacuum in the turbine exhaust region, with consequent increase in turbine power and efficiency.

The need for small droplets with high surface-volume ratio for higher heat transfer and evaporation rates has given rise to an additional research program at the Massachusetts Institute of Technology for the purpose of devising a suitable method for accurately measuring the drop sizes of water droplets moving in a high-velocity gas stream, and ultimately to determine by drop-size measurements what type of water injection nozzle and what dynamic and thermodynamic conditions will produce the smallest droplets.

At the present time the belief is being adhered to that droplet size under conditions existing in the Aerothermopressor is sensitive to the difference between inlet-gas velocity and water injection velocity, the higher relative velocity producing the smaller droplets.

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Aerothermopressor evaporation takes place as the water droplets travel downstream with the gas, with accompanying decrease in the temperature gradient between the two.

With an upper limit on turbine exhaust (Aerothermopressor inlet) stagnation temperature being imposed by turbine power requirements, the steady flow energy equation suggests that with both high gas velocity and high gas stream temperature being favorable to high evaporation rates, an optimum inlet velocity must exist for a given inlet stagnation temperature and water temperature.

It was the purpose of this thesis to determine experimentally whether this optimum velocity lies above or below the speed of sound for a typical turbine exhaust stagnation temperature of $1500^{\circ}R$.

Tests of a small-scale Aerothermopressor were conducted at inlet Mach numbers of approximately 1.5 and 1.35. The results were compared with subsonic data previously recorded for the same rig. From this comparison the conclusion was drawn that the optimum inlet velocity for the Aerothermopressor operating at $1500^{\circ}R$ lies in the supersonic range.

The main design feature incorporated in this thesis was to devise a means of converting a bellmouth nozzle to a supersonic nozzle. This was accomplished by the design of a water injection nozzle so shaped as to also function as an area plug to give an annular-shaped converging diverging nozzle.

Thesis Supervisor: Ascher H. Shapiro
Title: Professor of Mechanical Engineering

TO CHARLES WALKER MACKAY

Acknowledgment

I wish to thank Professor Ascher H. Shapiro, whose indomitable enthusiasm for all phases of scientific endeavor inspired me to overcome my own tendency to shy away from the laboratory side of the work and "round out" my academic career by undertaking an experimental thesis.

I wish also to thank Harry Foust, chief mechanic on the Aerothermopressor Project, whose voluntary participation in the operating phase of the experiment was equal to my own, and without whose help and guidance in the construction, assembly, and repair of the apparatus the operating phase would not yet have been entered.

To Associate Professor K.R. Wadleigh, Assistant Professor A.A. Fowle, and Mr. A.J. Erickson of the faculty and to Arthur Johnson and Donald Haradan of the Gas Turbine Laboratory, who were always ready to lend a hand when I found my back to the wall, I wish to express my appreciation for joining with Professor Shapiro and Harry Foust in making this venture a most pleasant experience.

Finally, I would add a word of thanks to Mary Kate, not only for bearing the lion's share of family and household responsibilities while I was occupied with this and associated work; but for her painstaking care in typing both the first draft and the final manuscript.

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I. Introduction

It is known (Reference 1) that one of the effects of cooling a gas stream is to raise its stagnation pressure. The "Aerothermopressor (Reference 2) was conceived by Professor A. H. Shapiro at the Massachusetts Institute of Technology to exploit this effect by injecting cooling water into a hot high-velocity gas stream. With the advent of the gas turbine as a prime mover for ship propulsion and power generation, interest has focused upon the Aerothermopressor (abbreviated "A.T.P.") as a means of improving gas turbine performance.

By receiving gas turbine exhaust gases, injecting cooling water into them, and exhausting to atmosphere the Aerothermopressor, with its attendant rise in stagnation pressure from inlet to exhaust, will provide the turbine with a back-pressure below atmospheric. Thus the turbine exhausts into a vacuum, with a consequent increase in its power and efficiency. The Aerothermopressor can be said to perform for the gas turbine cycle the same function as the condenser does for the steam turbine cycle, the only operating cost being the small amount of power required to pump sea water or river water into the Aerothermopressor at low velocity.

It appears from present knowledge of the phenomena occurring in the Aerothermopressor process that the best performance will eventually be achieved with either a high subsonic or low supersonic gas inlet velocity. The effect of inlet Mach number upon A.T.P. performance is discussed in

Section III.

The term "supersonic inlet" is utilized rather than the term "supersonic Aerothermopressor" since it has been found (Ref. 2) that in the subsonic-inlet A.T.P. process a continuous transition from subsonic to supersonic velocity is possible even in a constant-area duct. This phenomenon is caused by evaporation effects.

The experimental and theoretical research program now in progress at the Massachusetts Institute of Technology under the sponsorship of the Office of Naval Research has thus far concentrated upon subsonic gas inlet velocity, with its advantage of greater experimental flexibility and economy (variation of gas flow and inlet Mach number in the supersonic range ordinarily requires either a separate nozzle for each Mach number, or flexible walls in the nozzle region).

This thesis was undertaken for the purpose of obtaining some experimental data at "low" supersonic inlet velocities for comparison with data already recorded at high subsonic inlet velocities, to aid in answering the question as to whether optimum performance will demand an inlet gas velocity above or below the speed of sound. The only supersonic inlet A.T.P. data reported at the time of this writing was obtained at approximately Mach 2.0 on a 1.525" diameter constant-area A.T.P. without diffuser (Ref. 3).

At the time of this investigation a "medium" (11" dia-

meter test section) constant-area A.T.P. in the Gas Turbine Laboratory at the Massachusetts Institute of Technology was being used for subsonic-inlet experimental investigations of the effect of area variation upon performance, by means of conical area-plugs inserted inside the test section and moved longitudinally along the axis. This means of area-variation was chosen (with the disadvantage of added stagnation-pressure loss due to drag at the surface of the area plugs) over the more desirable possibility of varying the test section diameter along the length, due to greater flexibility and economy.

Prior to the development and installation of this "medium-scale" rig, Wadleigh in his doctoral dissertation (Ref. 4) investigated subsonic-inlet A.T.P. performance in a "small-scale" rig consisting of a 2 1/8" diameter constant-area test section supplied with hot gas from standard 6" pipe through an elliptical-contour bellmouth nozzle, and exhausting through a constant-angle diffuser into 6" standard pipe. Since Wadleigh's apparatus was available for use, and since the very nature of the Aerothermopressor process (high temperature plus corrosive gases) calls for stainless steel construction throughout, it was deemed advantageous from the point of view of economy as well as time to utilize this apparatus in the supersonic-inlet investigations.

At small diameters the predominance of wall friction effects (tending to reduce the stagnation pressure) preclude

the possibility of obtaining an actual rise in stagnation pressure by evaporative cooling, but as Wadleigh has pointed out, the measure of amount of reduction in stagnation pressure loss obtainable by introducing cooling in a small-scale rig may be used with some success as a criterion for estimating A.T.P. performance.

A complete summary of the experimental and theoretical work accomplished on the Aerothermopressor project to date (as well as a complete bibliography on the subject) is contained in Reference 2.

II. Main Test Equipment and Measurements

A schematic drawing of the main test equipment used in this investigation is given in Fig. 1. For reference purposes, all pressure taps are numbered in order, commencing at the upstream end of the apparatus. Two possible combinations were afforded by Wadleigh's apparatus: a constant-area test section 72" long in conjunction with a 6-degree diffuser, or a constant-area test section 36" long in conjunction with a 3-degree (total included angle) diffuser. Due to the problem of friction choking in a constant-area test section at low supersonic inlet Mach numbers, the shorter test section (2.125" I.D., 36" long) with the 3-degree diffuser (approximately 78" long) was chosen for the supersonic-inlet testing.

The test section was made from heavy-wall #321 stainless steel tubing and fitted with eleven static pressure taps .030" in diameter, spaced at intervals along the length (taps #2 to #12 in Fig. 1). The diffuser was rolled from #321 stainless steel sheet 1/8" thick and fitted with ten static pressure taps .030" in diameter spaced at intervals along the length (taps #13 to #22 in Fig. 1). The elliptical bellmouth was machined in a #321 stainless steel block 1 1/2" thick, welded to the inlet end of the test section. Table I tabulates the axial locations of all pressure taps and other pertinent stations along the duct length.

Hot combustion gases were supplied to the bellmouth through 6" stainless steel pipe by a natural gas ("city gas") furnace constructed by a the Etter Engineering Co., and utilizing an Eclipse NHE burner No. 5 as the primary heating burner and an Eclipse Walltite LEA 9 burner No. 3 as the pilot burner. Air was supplied to the furnace by a centrifugal blower. The gas-air mixture to the furnace was fixed by a pressure-regulator control system: hence good temperature control was achieved by merely controlling the air supply to the furnace with a large butterfly valve and a small bypass valve located in the air supply line between the blower and the pressure regulator.

The initial stagnation temerature T_{oi} of the gas was measured inside the standard 6" pipe at a location 9" from the entrance of the bellmouth, by means of a five shielded chromel-alumel thermocouple manufactured by the Aerotech Specialties Co. in Glastonbury, Conn., and read by a Leeds and Northrup K-2 potentiometer.

The initial stagnation pressure p_{oi} was measured by means of a stagnation pressure probe (tap #1 in Fig. 1) located in the 6" hot gas supply line 15" upstream from the bellmouth entrance.

Before exhausting the hot gases into the campus atmosphere, further cooling not necessary to the A.T.P. process was effected by discharging the diffuser through

a two-foot length of standard 6" pipe into a quench tank 7' high and 3' in diameter, fitted inside with two water sprays. Final stagnation pressure p_{o_f} was measured by means of a stagnation probe (tap #23 in Fig. 1) located in the 6" discharge pipe. To obtain and maintain gas flow through the test section, suction was provided at the top of the quench tank by means of a 6" steam ejector which is permanently installed in the Gas Turbine Laboratory. Back pressure on the diffuser exit was controlled by a 6" gate valve located between the 6" steam ejector and the quench tank. Excess quench water was pumped out of the bottom of the quench tank by adapting a small injector (which was readily available) to the job of an ejector.

All pressures were measured on a mercury manometer board as differences (cm. Hg.) from an atomspheric mercury column, and converted to absolute pressures by the local mercury barometer reading.

The problems of (1) furnishing the A.T.P. cooling-water supply at the entrance to the test section and (2) adapting Wadleigh's bellmouth nozzle to supersonic flow were solved simultaneously by designing one piece of equipment which for lack of a better name was called a "water injection nozzle assembly". Details of this assembly are shown in Figs. 2 and 3.

The main body of the water injection nozzle assembly

functions as an area plug which, when mounted inside the bellmouth and test section, yields an annular-shaped converging-diverging supersonic nozzle. The area variation of this plug was designed on a one-dimensional isentropic basis so as to fix the throat of the annular-shaped supersonic nozzle at the exit plane of the bellmouth nozzle, since a static pressure tap (tap #2) was located at this point.

In keeping with the current line of thinking in the Aerothermopressor Project as to means of water injection, the water injection nozzle supplies its water through six "atomizer tubes" of standard 12-gauge stainless steel tubing manufactured by the Hub Needle Company, Boston, Mass., discharging parallel to the direction of gas flow. These six atomizer tubes were spaced symmetrically about the test section axis at the root-mean-square radius of the test section so as to serve equal gas flow areas. Fig. 4 shows the water injection nozzle assembly mounted in place inside the bellmouth and test section to form the annular-shaped supersonic nozzle.

With this scheme, variation in inlet Mach number is obtainable by merely changing conical tips in the nozzle assembly. Two tips were actually manufactured and tested: conical tip No. 2 (Fig. 5) with diameter .840" at bellmouth exit plane (designed for Mach 1.5), and conical tip No. 3 (Fig. 6), with plug diameter .540" at bellmouth exit

plane (designed for Mach 1.35). Tip No. 1 was never manufactured, having been abandoned in favor of the smoother contours of tip No. 2.

The water injection nozzle assembly was manufactured of #304 stainless steel by the instrument room of Shop #31 at the U.S. Naval Shipyard, Boston, Mass.

Since the conical tip (and therefore the nozzle section) extends 4" into the 36" test section, the actual effective length of the constant-area test section for the supersonic runs is 32".

Water was supplied to the water injection nozzle assembly at city water pressure, and water flow was measure by a Schutte and Koerting rotameter (rotor No. 4B, tube No. 4R).

Gas flow through the choked converging-diverging nozzle was estimated by "Fliegner's formula" (with $k = 1.35$), modified by an estimated nozzle discharge coefficient $C_w = .98$. For this purpose the furnace combustion gases were assumed to have the same molecular weight and specific heats as air. The ratio of specific heats $k = 1.35$ was chosen as a mean value for the range of stream temperatures encountered in the two nozzles. For ready reference, isentropic flow functions for $k = 1.35$ are tabulated in Table II.

III. Effects of Inlet Mach Number on A.T.P. Performance

Based on a one-dimensional analysis utilizing the continuity, momentum, and energy equations for a perfect gas and neglecting the less influential effects of: (1) change in mass rate of gas flow due to "new" water vapor entering the gaseous phase by evaporation from the water droplets, (2) change in molecular weight of the gas due to the appearance of this same new water vapor, and (3) change in the ratio of specific heats k due to this new vapor and due to temperature change, the governing differential equation for the local stagnation pressure p_o of the gas at any cross-sectional plane through an A.T.P. duct without shock may be written (Ref. 2) :

$$dp_o = -\frac{kp_oM^2}{2}\left(\frac{dT_o}{T_o} + 4f\frac{dz}{D} + n\right)$$

In this equation M and T_o represent the local Mach number and stagnation temperature, z the axial distance along the duct from any fixed reference point (z increasing positive in the direction of the flow), D the local diameter, f the Fanning friction factor, and n a term representing droplet drag exerted on the gaseous phase. A more refined equation which includes the effects neglected here (or, for that matter, the governing differential equation for any of the other dependent flow variables) may be written down at once by referring to the Table of Influence Coefficients

in Ref. 2; but the above simple form is adequate for the purpose intended here. The above equation will be referred to henceforth as the fundamental governing equation of the A.T.P.

The factor enclosed in parentheses on the right-hand side of the fundamental equation represents the net influence of the major effects in the A.T.P. process: (1) evaporative cooling (decrease in gas T_o), tending to increase the stagnation pressure, and (2) wall friction(assisted by the initial water droplet drag), tending to decrease the stagnation pressure. In actual full size A.T.P. operation in which evaporative cooling effects predominate, the integrated form of the factor in parentheses will be positive, yielding a net rise in stagnation pressure from inlet to exhaust. Further, it may be seen from the factor M^2 that for such full-size operation the higher the Mach number level through the duct, the higher will be the stagnation pressure rise. For supersonic operation with shock, the loss in stagnation pressure across the shock must also be reckoned. In constant-area operation this shock influence must necessarily limit supersonic operation to the lower end of the supersonic range; with variable area in the duct (studied theoretically by Gavril, Ref. 7, and currently under experimental study with subsonic inlet by Assistant Professor A. A. Fowle) the possibility exists of diffusing

supersonic flow to a lower Mach number before shock, thus reducing shock losses.

The best A.T.P. performance does not, however, lie with the highest possible inlet Mach number as it might appear from this reliminary discussion, since the factor in parentheses in the above equation is also a function of Mach number. The term dT_o/T_o is the change in gas stagnation temperature brought about by evaporation of the water droplets, which in turn is greatly influenced by inlet Mach number, as discussed in the following paragraphs.

Water is believed to leave the "atomizer tubes" of the water injection nozzle in the form of "ligaments" or "sheets" which are broken up by the impact or drag of the high-velocity gas into water droplets (Refs. 5 and 7). It is further believed that the size of water droplets resulting from this process depends in large measure upon the relative velocity between the gas the the water--the higher the relative velocity the smaller the drops (Ref. 7). So important is drop size to A.T.P. performance that an additional research program is underway at the Massachusetts Institute of Technology on droplet technology, intended first to investigate several schemes of measuring drop size in a moving stream, and eventually to determine what conditions will yield the smallest drops. Information obtained from this program should be of value to combustion studies as well as to A.T.P. design.

For a fixed water-to-air ratio and a fixed initial

temperature difference between water droplets and gas, the smaller the droplet diameter the greater is the amount of water surface area exposed for heat transfer (by simple geometrical considerations) and hence the greater the evaporation rate. In addition to this effect of increased heat-transfer area, a smaller diameter means a lower Reynolds number; hence a higher coefficient of heat transfer. Gavril (Ref. 7) has demonstrated that due to the combination of these two effects the heat transfer rate between water droplets and gas varies inversely with roughly the square of the droplet diameter. Since evaporation due to heat transfer is the heart of the A.T.P. process, the importance of obtaining small water-droplet diameters cannot be overemphasized.

The evaporation must also depend upon the difference between gas stream temperature and water droplet temperature. During the evaporation process the water droplets are heated up and the gas is cooled; hence the temperature difference between the two is gradually reduced as the moist gas and water droplets move downstream. To maintain a high evaporation rate, then, a large initial temperature difference between air and water (at the water injection point) is desirable.

For a fixed initial stagnation temperature (such as the design turbine exhaust temperature of a gas turbine designed to be fitted with an Aerothermopressor) the gas may

be accelerated adiabatically to a high subsonic or low supersonic velocity prior to entering the A.T.P.; but (by the energy equation for steady flow) as the velocity is increased the stream temperature of the air is decreased. Thus the optimum inlet Mach number for the A.T.P. must achieve a balance between two contradictory requirements: high gas velocity to obtain small droplets from the water "breakup" process, and high gas stream temperature, both of which are favorable to high evaporation rates.

IV. Limitations Imposed by Test Section Length

Constant-area Aerothermopressor operation with supersonic inlet implies the existence of a shock somewhere in the duct. Since the purpose of the A.T.P. is to obtain a rise in stagnation pressure and since shock from high supersonic Mach number produces large losses in stagnation pressure, good supersonic-inlet A.T.P. operation at constant area must necessarily be limited to low supersonic Mach numbers with their attendant "weak" shocks.

For this reason it was considered desirable to investigate supersonic Mach numbers of about 1.5 and below. However, the length of the test section (length L = 32", diameter D = 2.125", L/D = 15) imposed a minimum value on the supersonic Mach numbers which could be achieved.

For a given constant-area test section with Fanning friction f and a length-diameter ratio L/D, there is in "dry" gas flow a maximum subsonic and minimum supersonic Mach number which can exist at the test section entrance, corresponding to friction choke at the test section exit. (Ref. 8) These values are tabulated in the "Fanno Line" tables (eg. ref. 6). With the 15-diameter length of test section employed in these tests assuming, for instance, a friction factor f = .005, the computed value 4fL/D = 0.3 yields (for k = 1.4) values of $M_{subsonic_{max}} = 0.66$ and $M_{supersonic_{min}} = 1.98$.

Thus it was not to be expected that dry supersonic flow could be achieved in the existing test-section at Mach numbers 1.50 and 1.35, and dry supersonic flow was in fact found experimentally to be impossible. (Fig. 9).

The theoretical one-dimensional analyses which have been carried out (Refs. 4 and 7) have resulted in a "Table of Influence Coefficients" (see also Ref. 2) which show the effect of a change in any one of the arbitrarily chosen independent properties upon each of the remaining dependent properties associated with A.T.P. flow. Use of these coefficients in interpreting Wadleigh's experimental data and Gavril's numerical computations (carried out on the "Whirlwind" high-speed electronic digital computer at Massachusetts Institute of Technology) has resulted in Table III (reproduced from Ref. 2) which shows the effects of the four major controlling parameters of the Aerothermopressor (area change, evaporation, wall friction, and liquid acceleration) upon the flow properties.

From Table III it may be seen that the effect of wall friction in the A.T.P. is to "drive" the Mach number toward unity (just as in dry Fanno-type flow), while the effect of evaporation is to drive the Mach number away from unity. Knowing these two effects to be opposite to each other, it was expected (and actually realized experimentally) that "wet" A.T.P. supersonic flow might be obtained at supersonic Mach numbers lower than the limiting

value imposed by friction choke in the dry flow. Conical tip No. 2 (Mach 1.5) was tried first and found to provide wet supersonic flow in the test section. After this was found successful, a still lower Mach number of 1.35 (conical tip No. 3) was tried and found also to be successful. Mach number 1.35 is believed to be very close to the minimum supersonic inlet Mach number for which supersonic-inlet A.T.P. operation can be achieved with the present test section since at this Mach number supersonic flow was obtainable only at a very critical value of water air ratio. Any slight increase or decrease in the water flow from this level resulted in unstable flow which quickly resulted in steady subsonic flow with friction choke at the test section exit. No attempt was therefore made to go to supersonic Mach numbers below 1.35.

In his subsonic-inlet runs Wadleigh was able to estimate nozzle losses and Fanning friction factor by "hot dry" and "cold dry" data. This was not possible for the supersonic-inlet case. since dry supersonic runs were prohibited by the friction-choke phenomenon just mentioned. However, at Wadleigh's suggestion, with the water supply to the Water Injection Nozzle cut off, auxiliary water was injected radially into the stream through six symmetrically-spaced taps in the test-section wall located 6" downstream from the nozzle exit (midway between pressure taps #5 and #6).

The evaporative effects of this auxiliary water were sufficient in the case of the Mach 1.5 nozzle to eliminate the friction choke at the test section exit and produce supersonic flow in the entire test section. This gave "dry" supersonic pressure data as far downstream as tap #5 (Run #34, Fig. 9), from which actual flow Mach No. and nozzle losses were estimated through the well-known formulas of "dry" gas dynamics (Section VI). In the case of the Mach 1.35 nozzle, this radial water injection was not sufficient to overcome the friction choke; hence dry supersonic data for this nozzle is not available.

V. Experimental Results

For the purpose of comparing Aerothermopressor performance at low supersonic entry with performance at subsonic entry, Aerothermopressor runs (runs with water injection at test-section inlet) were made at nominal inlet Mach numbers 1.5 and 1.35, utilizing conical tips No. 2 and No. 3. An initial stagnation temperature $T_{o_i} = 1500^{\circ}R$ was chosen for this comparison for two reasons: (1) it represents a reasonable exhaust temperature for an open-cycle gas turbine and (2) considerable subsonic-entry data was available at this temperature for comparison purposes.

Curves of static pressure vs. axial distance along the duct for these runs are plotted in Figs. 7 and 8. Since the velocity at the stagnation pressure taps #1 and #23 (inside the 6" pipe) is negligible, the stagnation pressures measured at these two stations also represent (within the accuracy of the mearsurements themselves) the static pressures. Since the walls of the furnace were partially open to atmosphere, the initial stagnation pressure p_{o_i} at tap #1 was essentially atmospheric in all the runs. For convenience, the pressures are plotted in the dimensionless ratio form p/p_{o_i}, and a schematic of the duct (showing locations of the various pressure taps by reference number) is drawn directly on the graphs.

Fig. 9 presents the dry characteristics (no water

supplied to the stream) of the apparatus with the two different nozzles, showing for the Mach 1.5 tip a shock in the supersonic nozzle followed by subsonic friction choke at the test-section exit, and for the Mach 1.35 nozzle a purely subsonic flow with friction choke at the test-section exit. As mentioned previously, Fig. 9 also shows the dry supersonic nozzle pressure data obtained for the Mach 1.5 nozzle by downstream auxiliary radial injection.

As mentioned in Section IV, the water-air ratio required to maintain wet supersonic flow with the Mach 1.35 nozzle was extremely critical; hence water-air ratio was not available as a variable. With the Mach 1.5 nozzle, the band of water-air ratios for which stable supersonic flow could be maintained was slightly wider. With this nozzle runs were made at approximately the maximum and minimum water-air ratios permissible for supersonic flow, but even here the band was still so narrow that practically no effect on performance could be imposed by varying the water supply.

For the runs in Figs. 7 and 8 the water supply valve was adjusted to a point midway between the two cut-off points (too much water and too little water) at which the supersonic flow became unstable. The water-air ratios for this condition of operation were .235 for Mach 1.5 and .169 for Mach 1.35, which may be considered to be "optimum" values for the particular apparatus used.

Before 1500°R was selected for the supersonic-subsonic comparisons, the initial stagnation temperature T_{o_i} was varied between the minimum temperature required for stable operation and a maximum of 1800°R. Within this range of temperatures the effect of T_{o_i} on A.T.P. performance was very slight--the variation in pressure differences obtainable by varying T_{o_i} being measured in fractions of a centimeter of mercury.

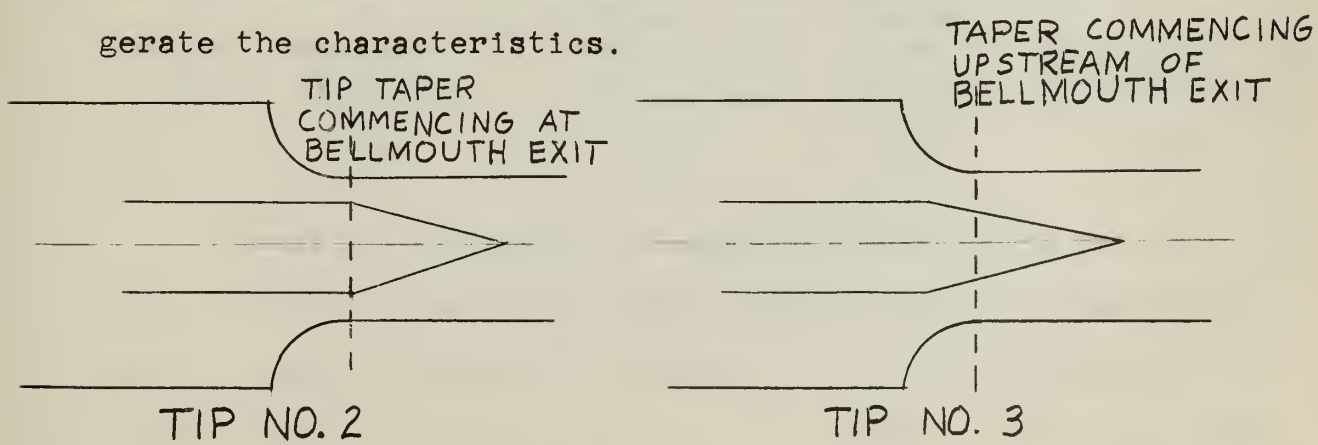
In interpreting the plots of Figs. 7,8, and 9 it should be borne in mind that the curves were constructed by simply connecting with straight-line segments the pressure values which were measured at intervals along the duct length. For this reason the peaks and valleys in these plots do not necessarily represent the maximum and minimum pressures in the actual flow--these may just as likely have occurred between pressure taps.

VI. Determination of Inlet Mach Number and Nozzle Losses

Although conical tips No. 2 and No. 3 are referred to throughout this report as providing "Mach 1.5" and "Mach 1.35", respectively, these are merely nominal Mach number values. The geometric area ratios for the annular nozzles formed by these tips are representable in a one-dimensionable flow analysis by:

$$\frac{A_{test}}{A_{throat}} = \frac{D^2_{test}}{D^2_{test} - D^2_{tip_{crit.}} - 6\,D^2_{tube}}$$

where D_{test} is the diameter of the test section (2.125"), $D_{tip_{crit}}$ is the "critical" tip diameter at the cross-section lying in the plane of the bellmouth exit (.840" for tip No. 2: .540" for tip No. 3), and D_{tube} is the outside diameter of the atomizer tubing (.109"). Since the diameter of the Water Supply Plug (Fig. 2, part A) is .840", the annular nozzles formed by tips No. 2 and No. 3 are of the character illustrated in the sketches below, where the elliptical contour has been replaced by a circular contour and the tips are represented by large-angle right-circular cones to exaggerate the characteristics.



From these sketches it can be seen that while the geometric throat formed by tip No 2 is theoretically located exactly at the bellmouth exit, the throat formed by tip No 3 is actually slightly upstream of the bellmouth exit; hence the foregoing representation for nozzle area ratio is somewhat less accurate for tip No.3 than for tip No 2. This displacement is probably not great enough to introduce any serious error in the computed value of area ratio in this nozzle; but it must be borne in mind in interpreting the curves of Fig. 8 that tap #2 does not record the throat pressure for this nozzle. This accounts for the peculiar characteristic of the pressure curves between taps #2 and #3 in Fig. 8.

Based upon this method of computing area ratio, the area data and the corresponding values of Mach number M from the isentropic flow functions of Table II are tabulated below for the two nozzles. The area of the test section is 3.547 squares inches.

Conical Tip	Nominal Mach No.	Nozzle Throat Area (sq. in.)	Nozzle Area Ratio	Corresponding M (isentropic)
No. 2	1.5	2.936	1.208	1.53
No. 3	1.35	3.263	1.087	1.34

With the dry nozzle pressure data obtained for the Mach 1.5 tip by auxiliary radial water injection downstream of the nozzle (Section IV and Fig. 9), a very convenient method (Ref. 8) is available for calculating the nozzle exit

Mach number, utilizing an estimated nozzle discharge coefficient C_w:

$$\left[\left(\frac{p}{p_o}\right)\left(\frac{A}{A^*}\right)\right]_{isen} = \frac{1}{C_w}\left(\frac{p}{p_{o_i}}\right)_{meas.}\left(\frac{A_{test}}{A_{throat}}\right)_{geom.}$$

In this relationship the quantity on the left represents a commonly tabulated (e.g., Ref. 6) isentropic flow function. For easy reference, this function is tabulated in Table II for k = 1.35. With the measured nozzle exit pressure ratio p/p_{o_i} = .274 (tap #4, run #34) and the above-recorded geometric area ratio:

$$\left[\left(\frac{p}{p_o}\right)\left(\frac{A}{A^*}\right)\right]_{isen} = \frac{.274 \times 1.208}{.98} = .338$$

From Table II this gives a nozzle exit (test-section entrance) Mach number of 1.47 for conical tip No. 2.

Taking this value of 1.47 to be the best available estimate of the actual flow Mach No. at nozzle exit, an estimate may be made of the nozzle losses. Denoting nozzle exit conditions by the subscript "e":

M_e = 1.47 gives p_e/p_{o_e} = .290 (From Table II)

p_e/p_{o_i} = .274 (measured, tap #4, run #34)

$$p_{o_e}/p_{o_i} = \frac{p_e/p_{o_i}}{p_e/p_{o_e}} = .274/.290 = .945$$

It should be noted that this loss parameter is based upon an assumed value of .98 for the nozzle discharge coef-

ficient.

The rise in stagnation pressure exhibited in the dry constant-area supersonic flow between taps #4 and #5 in run #34 is attributed to wall friction (Table III).

VII. Mass Flow of Air

Following the terminology which has come into usage in the Aerothermopressor Project, the hot dry furnace gases entering the A.T.P. will be referred to henceforth in this report as "air", the term "gas" being reserved for the gaseous phase of the wet mixture of combustion gases, water vapor, and water droplets which exists in the duct after water injection.

Mass flow of air was calculated from Fliegner's formula (Ref. 8) for a choked nozzle, modified by a nozzle discharge coefficient C_w:

$$w_a = C_w \sqrt{\frac{g_o k}{R}\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}\left(\frac{p_{o_i} A_{throat}}{\sqrt{T_o}}\right)$$

Taking $k = 1.35$, $R = 53.35 \frac{\text{ft-lbf}}{\text{lbm-}^\circ\text{R}}$, $g_o = 32.17 \frac{\text{lbm-ft}}{\text{lbf-sec}^2}$ and $C_w = .98$, this has the form:

$$w_a = .515 \frac{p_{o_i} A_{throat}}{\sqrt{T_{o_i}}}$$

At the measured operating values of $p_{o_i} = 14.5$ psia and $T_{o_i} = 1500^\circ$R, this gives for the air flow passed by the two nozzles:

Conical Tip	Air Mass Flow w_a (lb./sec.)
No. 2 (Mach 1.5)	.565
No. 3 (Mach 1.35)	.629

VIII. Interpretation of the Static Pressure Curves

The three major regimes (Ref. 2) of constant-area A.T.P. behavior are clearly visible in run #32 (Fig. 7) and run #44 (Fig. 8).

In regime I droplet drag is predominant. Initial droplet drag (liquid acceleration) in supersonic flow tends to increase the static pressure (Table III). This regime is illustrated between taps No. 4 and #5 in run #32 and between taps #3 and #4 in run #44.

In regime II evaporation is predominant. Evaporation in supersonic flow tends to decrease the static pressure (Table III). This regime is illustrated between taps #5 and #10 in run #32 and between taps #4 and #9 in run #44.

In regime III the difference between water droplet temperature and gas stream temperature has decreased to the point where evaporation is no longer controlling, and wall friction becomes predominant, causing (Table III) an increase in static pressure. This is illustrated between taps #11 and #12 in run #32 and between taps #9 and #12 in run #44.

The remaining portion of the curves in these two runs represent shock to subsonic flow with its accompanying pressure rise, followed by pressure increase induced by area increase in subsonic flow.

The other runs in Figs. 7 and 8 represent operation with the shock in different locations, the shock patterns being clearly visible on the graphs. In runs #33 and #45 the shock is in the diffuser, with the drop in static pressure between taps #12 and #13 being induced by the area increase in the diffuser.

In run #26 (Fig. 7) the shock lies in the nozzle section. In this run regimes II and III take place in subsonic flow; regime I takes place in or near the shock zone. The static pressure rise between taps #4 and #10 is due to evaporation: the drop between taps #10 and #12 is due to wall friction.

IX. Performance Comparisons

A. Coefficient of Over-all Performance

It will be found useful to write the fundamental equation of the A.T.P. (Section III) in "normalized" form by dividing through by the product of initial stagnation pressure and the square of the inlet Mach number:

$$\frac{dp_o}{p_{o_i} M^2_{inlet}} = -\frac{k}{2}\left(\frac{p_o}{p_{o_i}}\right)\left(\frac{M}{M_{inlet}}\right)^2\left(\frac{dT_o}{T_o} + 4f\frac{dz}{D} + \eta\right)$$

In the search for the optimum inlet Mach number for the A.T.P. the basic consideration is the proper balance between the contradictory requirements of high initial relative velocity and high initial temperature difference between the air and water at the injection point. Downstream of the injection point the local Mach number and therefore the gas stream temperature may be controlled by area variation, as pointed out by Gavril (Ref. 7). Area variation used with moderation so as not to induce losses due to boundary layer phenomena has no effect upon stagnation pressure (Table III). Controlled area variation in the design of a full-scale Aerothermopressor involves two contradictory requirements: high Mach number level through the duct (the factor M^2 in the fundamental equation), and high temperature difference between gas and water throughout the evaporative section.

The answer to the question of balance between the

requirements of high initial relative velocity and high initial temperature difference on the basis of small scale conatant-area experimental data lies in the magnitude of the last factor on the right-hand side of the fundamental equation, rather than in the net over-all rise (or loss) in stagnation pressure, since the net over-all change in stagnation pressure can be seen to be strongly influenced by Mach number level throughout the length of the duct.

By visualizing an integrated form of the normalized fundamental equation, it may be seen that for runs made in the same apparatus and having essentially the same p_o profiles and M profiles (i.e., runs for which plots of p_o/p_{o_i} vs. axial distance z along the duct would lie essentially along one and the same curve, and similarly for plots of M/M_{inlet} vs. z), the relative magnitudes of the quantity $(p_{o_f} - p_{o_i})/p_{o_i}M^2_{inlet}$ or its equivalent $\left[\frac{(p_{o_f}/p_{o_i})-1}{M^2_{inlet}}\right]$ should yield the relative magnitudes of the net effect of evaporation vs. friction over the duct length, as influenced by initial conditions. This quantity has in fact been commonly used on the A.T.P. Project as a coefficient of over-all performance.

B. Supersonic Inlet vs. Subsonic Inlet

Due to the inherent differences in the general flow character of the supersonic-inlet and subsonic-inlet constant-area Aerothermopressors (opposite effects of evapo-

ration and friction on Mach number, plus the presence of shock in one but not the other), comparison between a subsonic-inlet run and a supersonic-inlet run by means of the over-all coefficient of performance defined above must be carried out with caution.

In order to eliminate as many variables as possible from the problem, comparison of the supersonic data was made with some previously unreported subsonic data recorded by Mr. A.J. Erickson of the faculty at the Massachusetts Institute of Technology using the same test section and diffuser, rather than with the published data of Wadleigh (Ref. 4) which was taken on a 72" test section and 6° diffuser.

Among Erickson's data are runs at 1500°R for inlet Mach numbers of 0.5 and 0.65 with different water-air ratios, utilizing Wadleigh's water injection nozzle No.3 (Ref. 4) which injected water at the plane of the bell-mouth exit. His 1500°R run with best coefficient of over-all performance (run B-1-b-4 by his designation system) is plotted in Fig. 10.

As previously pointed out, run #26 (Fig. 7) represents supersonic-inlet with shock in the nozzle region followed by subsonic flow throughout the test section and diffuser, with consequent p_o and M profiles essentially similar to Erickson's run in Fig. 10. For this reason

run #26 was chosen for the comparison. Its performance coefficient was computed utilizing a value of 1.47 for inlet Mach number, based on area ratio and measured pressure (Section VI). Performance coefficients for these two runs are tabulated below:

Run	M_{inlet}	p_{o_f}/p_{o_i}	$\left[\frac{(p_{o_f}/p_{o_i}) - 1}{M^2_{inlet}}\right]$
MacKay #26	1.47	.868	-.061
Erickson #B-1-b-4	0.65	.965	-.083

Since supersonic-inlet shows a substantially better coefficient of performance even while reflecting a shock loss, it appears that for an initial stagnation temperature of 1500°, better evaporation rates are definitely obtainable at supersonic inlet Mach numbers. Certainly serious consideration of supersonic-inlet in the design of a large-scale A.T.P. seems warranted, especially with the possibility of using area variation to diffuse the supersonic flow to a lower Mach number before shock (having at the same time the effect of increasing the temperature differential between gas and water).

C. Mach 1.5 vs. Mach 1.35

In utilizing the performance coefficient of IX-A to compare Mach 1.5 with Mach 1.35, runs #27 and #39 were chosen, since these have shock regimes commencing at approxi-

nately the same point in the duct, and hence have somewhat similar p_o and M profiles.

In the absence of dry nozzle pressure data for conical tip No. 3, both performance coefficients for this comparison were computed using inlet Mach numbers corresponding to area ratio alone, as tabulated in Section VI. On this basis the performance data was as follows:

Run	M_{inlet}	p_{o_f}/p_{o_i}	$\left[\frac{(p_{o_f}/p_{o_i}) - 1}{M^2_{inlet}}\right]$
#27	1.53	.850	-.064
#39	1.34	.881	-.066

Since these two Mach numbers yield essentially equal performance coefficients while reflecting unequal shock losses, it seems reasonable to assume that the optimum inlet Mach number for an A.T.P. with controlled area variation may conceivably lie higher than 1.47 (best estimate of flow Mach with tip No. 2) for inlet stagnation temperature of the order of 1500°R.

X. Suggestion for Further Study

Because of the rather crude estimate of Mach numbers used in comparing Mach 1.5 with Mach 1.35 (nominal), no decision made on the basis of the computations in Section IX-C can be considered decisive. For a more accurate determination of the optimum supersonic inlet Mach number it would be advisable in any further small-scale testing to shorten the test section in order to obtain dry nozzle pressure data for inlet Mach numbers below 1.47. From runs #26. #33, and #45 (Figs. 7 and 8) it is evident that little would be lost by chopping off the present test section at tap No. 10, since at this point the undesirable influence of wall friction "takes over" from evaporation and becomes controlling.

Shortening the test section by this amount would, with the present water injection nozzle length, leave an effective constant area duct length of 24" extending from tap #4 (supersonic nozzle exit) to tap #10. With an assumed $f = .005$ this gives $4fL/D = .226$ and a value of $M_{supersonic\ min}$ in the neighorhood of 1.7. With this shorter test-section length and its corresponding reduction of minimum supersonic inlet Mach number for no friction choke from about 1.98 (Section IV) to about 1.7, it is possible (1) that dry supersonic nozzle data might be obtainable for conical tip No. 3 by auxiliary radial water

injection downstream, and (2) a run for the Mach 1.35 nozzle might be obtained having the same character as run #26 for the Mach 1.5 nozzle. With the apparatus used in this experiment it was impossible with tip No. 3 to drive the shock any further upstream than run #38 (Fig. 8). since an increase in back pressure from this level resulted in subsonic nozzle flow.

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2. "The Aerothermopressor--a Device for Improving the Performance of a Gas Turbine Power Plant", by Ascher H. Shapiro, K. R. Wadleigh, B. D. Gavril, and A. A. Fowle, Mass. Inst. of Tech., prepared for future publication.

3. "Small Scale Constant-Area Test of an Aerothermopressor", by P. A. Gisvold, Jr. and J. C. Matheson, thesis for Nav. Eng. degree, M.I.T. (1952).

4. "An Experimental Investigation of a Small Scale Aerothermopressor--a Device for Increasing the Stagnation Pressure of a High-Temperature, High-Velocity Gas Stream by Evaporative Cooling", by K.R. Wadleigh, thesis for Sc. D. degree in Mech. Eng., M.I.T. (1953)

5. A Theoretical Investigation of the Heating-Up Period of Injected Fuel Droplets Vaporizing in Air", by M.M. El Wakil, O.A. Uyehara, and P.S. Myers, Univ. of Wisc., NACA TN 3179, 1954

6. "Gas Tables", by J.H. Keenan and J. Kaye, New York, John Wiley and Sons, 1945

7. "A Theoretical Investigation of the Thermodynamic and Dynamic Effects of Water Injection into High-Velocity, High-Temperature Gas Streams", by B.D. Gavril, thesis for Sc. D. degree in Mech. Eng., M.I.T. (1954)

8. "The Dynamics and Thermodynamics of Compressible Fluid Flow", by Ascher H. Shapiro, New York, the Ronald Press Co., 1954, vol. 1

TABLE I

LOCATION OF PERTINENT STATIONS ALONG THE DUCT

Station	Distance Downstream From Previous Station (inches)	Distance From Bellmouth Entrance (In.)
Stag. Pres. Probe #1	0	-15*
Bellmouth Entrance	15	0
Nozzle Throat (Tap #2)	2 1/8	2 1/8
Tap #3	2	4 1/8
A.T.P. Water Injection	1 5/8	5 3/4
Nozzle Exit (tap #4)	3/8	6 1/8
Tap #5	4	10 1/8
Auxiliary Water Injection	2	12 1/8
Tap #6	2	14 1/8
Tap #7	4	18 1/8
Tap #8	4	22 1/8
Tap #9	4	26 1/8
Tap #10	4	30 1/8
Tap #11	4	34 1/8
Tap #12	2	36 1/8
Diffuser entrance	2	38 1/8
Tap #13	2 7/8	41
Tap #14	8	49
Tap #15	8	57
Tap #16	8	65
Tap #17	8	73
Tap #18	8	81
Tap #19	8	89
Tap #20	8	97
Tap #21	8	105
Tap #22	8	113
Diffuser Exit	2 1/2	115 1/2
Stag. Pres. Probe #23	9	124 1/2
Quench tank entrance	19	143 1/2

* Indicates upstream

TABLE II

ISENTROPIC FLOW FUNCTIONS, k = 1.35

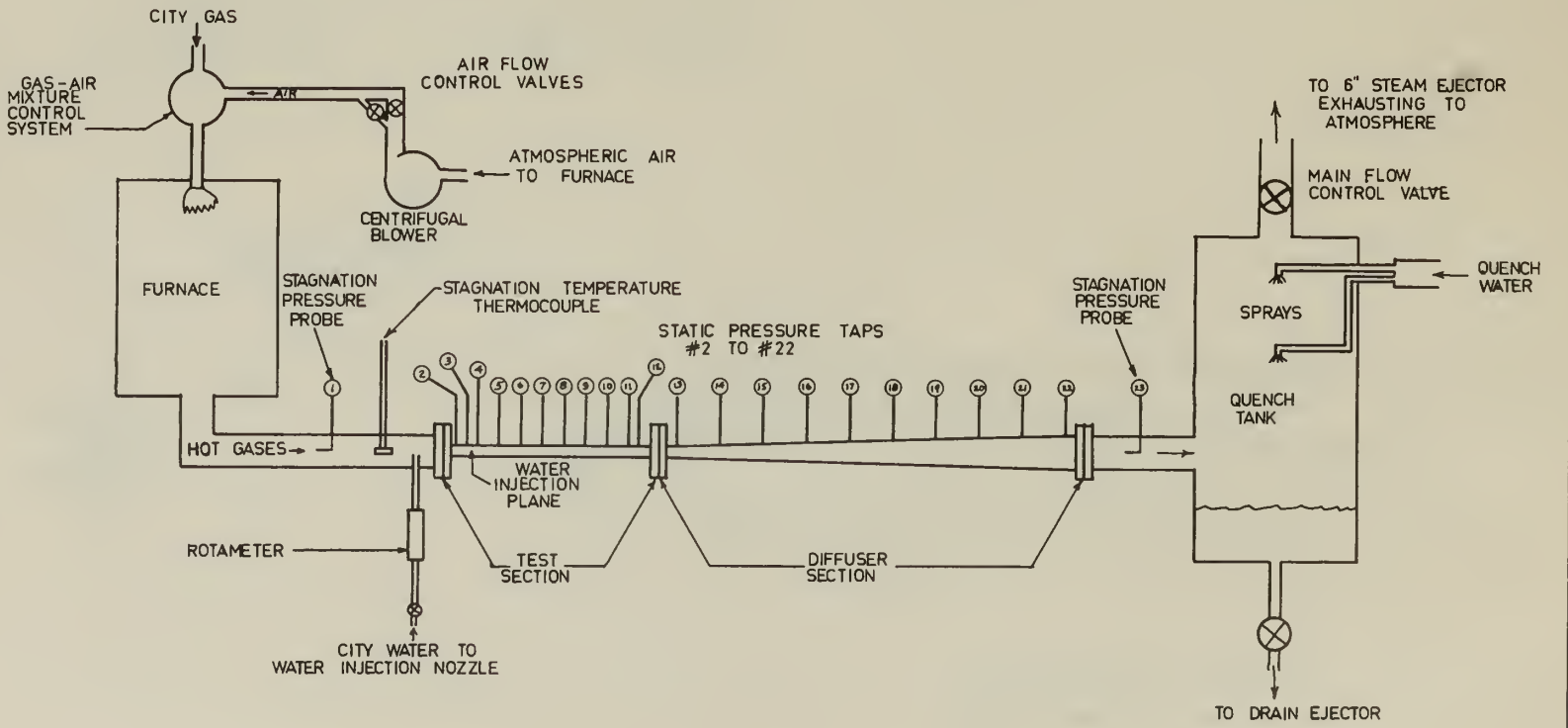
M	T/T_o	p/p_o	A/A^*	pA/p_oA^*
1.00	.851	.537	1.000	.537
1.29	.775	.373	1.064	.397
1.30	.772	.368	1.069	.393
1.31	.769	.364	1.072	.390
1.32	.766	.358	1.078	.386
1.33	.763	.353	1.083	.382
1.34	.761	.349	1.086	.379
1.35	.758	.344	1.092	.376
1.36	.755	.339	1.098	.372
1.37	.753	.335	1.101	.369
1.43	.736	.307	1.137	.349
1.44	.734	.303	1.143	.346
1.45	.731	.299	1.150	.344
1.46	.728	.294	1.155	.340
1.47	.726	.290	1.162	.337
1.48	.723	.286	1.168	.334
1.49	.720	.282	1.174	.331
1.50	.717	.278	1.183	.329
1.51	.715	.274	1.190	.326
1.52	.712	.270	1.196	.323
1.53	.709	.265	1.206	.320
1.54	.707	.262	1.212	.318
1.55	.704	.259	1.219	.316
1.56	.701	.254	1.228	.312

TABLE III

BEHAVIOR OF STREAM PROPERTIES UNDER INFLUENCE OF AREA CHANGE, EVAPORATION, WALL FRICTION, AND DROPLET DRAG

		Area increase produces (a)	Evaporation produces (b)	Wall friction produces (c)	Liquid acceleration produces (d)
Mach Number M	subsonic	decrease	decrease(h)	increase	increase (e)
	supersonic	increase	increase(h)	decrease	decrease (e)
Gas velocity V	subsonic	decrease	decrease(h)	increase	increase(e)
	supersonic	increase	increase(h)	decrease	decrease(e)
Pressure p	subsonic	increase	increase(h)	decrease	decrease(e)
	supersonic	decrease	decrease(h)	increase	increase(e)
Temperature T	subsonic	increase	decrease(h)	decrease	decrease(e)
	supersonic	decrease	increase(h)	increase	increase(e)
Gas Stagnation Temperature T_o	subsonic	nil	decrease	nil	decrease(f)
	supersonic	nil	decrease	nil	decrease(f)
Mixture Stagnation Temp. θ_o	subsonic	nil	decrease	nil	nil(g)
	supersonic	nil	decrease	nil	nil(g)
Gas stagnation Pressure p_o	subsonic	nil	increase(h)	decrease	decrease(e)
	supersonic	nil	increase(h)	decrease	decrease(e)
Mixture Stagnation Pressure P_o	subsonic	nil	increase(h)	decrease	decrease(g)
	supersonic	nil	increase(h)	decrease	decrease(g)

NOTES:
(a) Opposite effects for area decrease
(b) Opposite effects for condensation
(c) Opposite effects are impossible
(d) When $y<1$, $dV_\ell>0$; when $y>1$, $dV_\ell<0$ (See Ref. 2)
(e) Dependent upon magnitude of y for liquid deceleration (See Ref. 2)
(f) Opposite effect for liquid deceleration
(g) Same effect for liquid deceleration
(h) Based on β only, and generally correct for β in excess of two; otherwise effects are indeterminate (See Ref. 2)

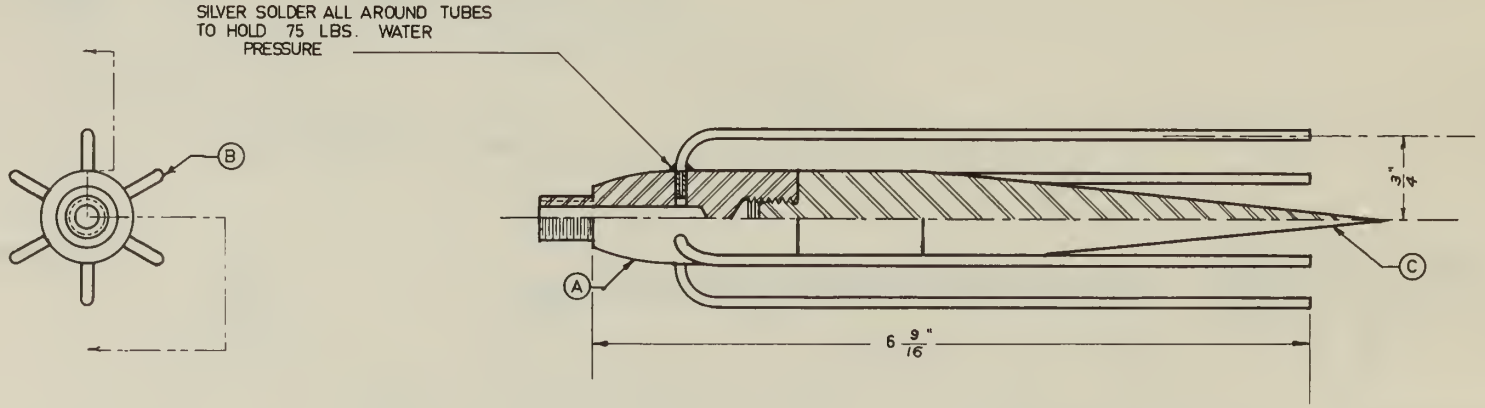


SCHEMATIC LAYOUT OF MAIN TEST EQUIPMENT

FIG. 1

FIG 1

PART	NAME	NO. REQ'D	MATERIAL	DWG. NO.
A	WATER SUPPLY PLUG	1	18-8 S.S.	RTM-D1
B	ATOMIZER TUBE	6	18-8 S.S.	RTM-D1
C	CONICAL TIP NO. 1	1	18-8 S.S.	RTM-D1

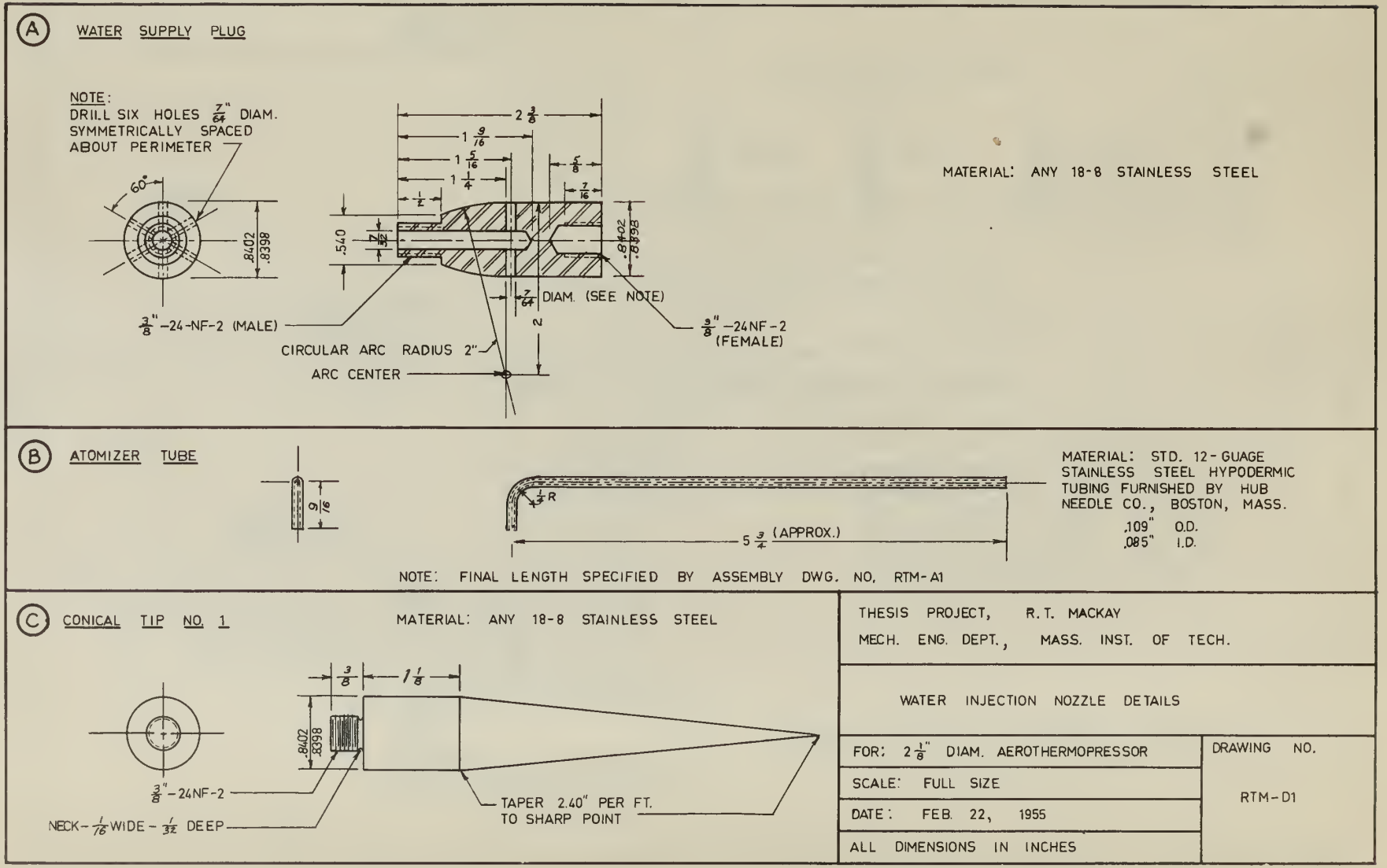


THESIS PROJECT, R. T. MACKAY MECH. ENG. DEPT., MASS. INST. OF TECH.	
WATER INJECTION NOZZLE ASSEMBLY	
FOR: 2 1/8" DIAM. AEROTHERMOPRESSOR	
SCALE: FULL SIZE	DRAWING NO.
DATE: FEB. 22, 1955	RTM-A1
ALL DIMENSIONS IN INCHES	

FIG 2

FIG. 2

FIG 3



THESIS PROJECT, R. T. MACKAY
MECH. ENG. DEPT., MASS. INST. OF TECH.

WATER INJECTION NOZZLE DETAILS

FOR: 2 1/8" DIAM. AEROTHERMOPRESSOR	DRAWING NO.
SCALE: FULL SIZE	RTM-D1
DATE: FEB. 22, 1955	
ALL DIMENSIONS IN INCHES	

FIG. 3

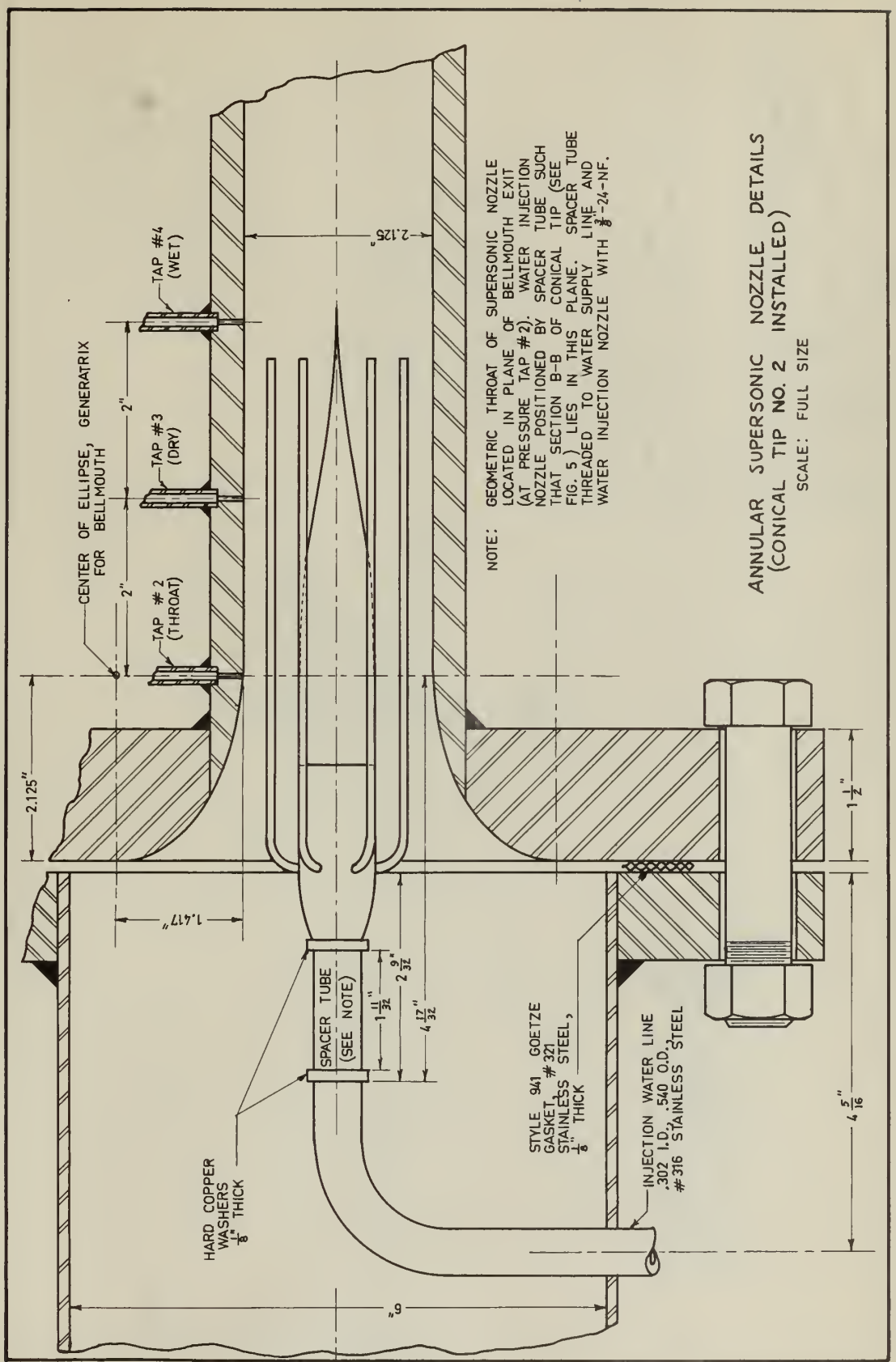
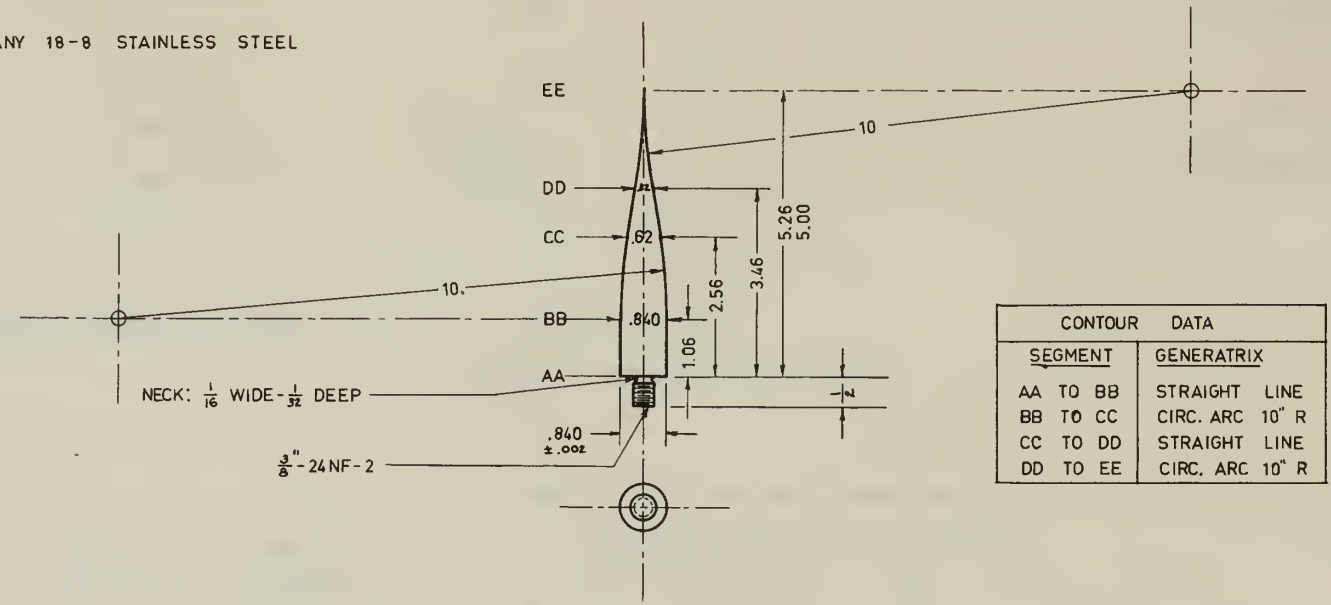

ANNULAR SUPERSONIC NOZZLE DETAILS
(CONICAL TIP NO. 2 INSTALLED)
SCALE: FULL SIZE
NOTE: GEOMETRIC THROAT OF SUPERSONIC NOZZLE LOCATED IN PLANE OF BELLMOUTH EXIT (AT PRESSURE TAP #2). WATER INJECTION NOZZLE POSITIONED BY SPACER TUBE SUCH THAT SECTION B-B OF CONICAL TIP (SEE FIG. 5) LIES IN THIS PLANE. SPACER TUBE THREADED TO WATER SUPPLY LINE AND WATER INJECTION NOZZLE WITH 3/8"-24-NF.
TAP #4 (WET)
TAP #3 (DRY)
TAP #2 (THROAT)
CENTER OF ELLIPSE, GENERATRIX FOR BELLMOUTH
2"
2"
2.125"
2.125"
1.417"
SPACER TUBE (SEE NOTE)
1 11/32"
2 9/32"
4 17/32"
STYLE 941 GOETZE GASKET #321 STAINLESS STEEL, 1/8" THICK
HARD COPPER WASHERS 1/8" THICK
INJECTION WATER LINE .302 I.D. .540 O.D. #316 STAINLESS STEEL
1 1/2"
4 5/16"
6"

FIG 4

FIG. 4

MATERIAL: ANY 18-8 STAINLESS STEEL



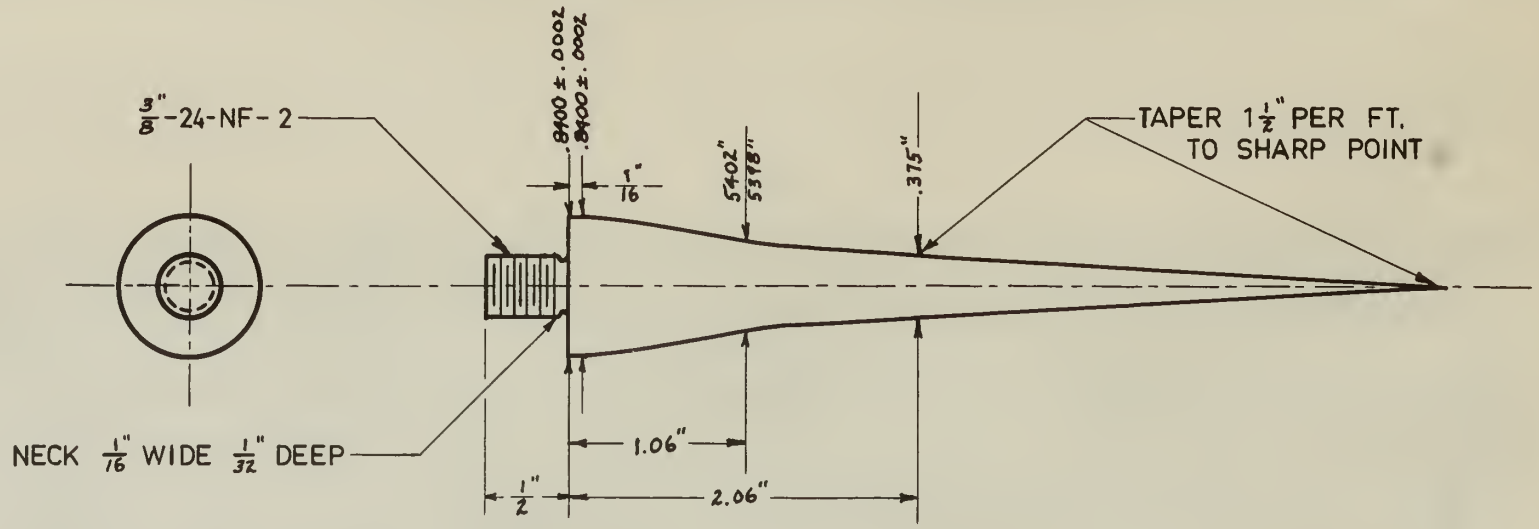
CONTOUR DATA	
SEGMENT	GENERATRIX
AA TO BB	STRAIGHT LINE
BB TO CC	CIRC. ARC 10" R
CC TO DD	STRAIGHT LINE
DD TO EE	CIRC. ARC 10" R

NOTE: FULL-SIZE TEMPLATE FURNISHED SEPARATELY. TEMPLATE PROVIDES FOR ZERO DIAMETER AT SECTION EE. MACHINE USING TEMPLATE TO SMALLEST PRACTICABLE DIAMETER, THEN HONE REMAINDER OF TIP TO CONCAVE NEEDLE-SHARP POINT APPROX. AS SHOWN. NOTE LIMITS ON TIP LENGTH.

THESIS PROJECT, R. T. MACKAY MECH. ENG. DEPT., MASS. INST. OF TECH.	
CONICAL TIP NO. 2	
FOR: WATER INJECTION NOZZLE, 2 1/8" AEROTHERMOPRESSOR	
SCALE: HALF SIZE	DRAWING NO.
DATE: FEB. 24, 1955	RTM-D2
ALL DIMENSIONS IN INCHES	

FIG. 5

FIG. 5



NOTE: FULL-SIZE TEMPLATE FURNISHED SEPARATELY

THESIS PROJECT, R. T. MACKAY MECH. ENG. DEPT., MASS. INST. OF TECH.	
CONICAL TIP NO. 3	
FOR: WATER INJECTION NOZZLE, 2 1/8" DIAM. AEROTHERMOPRESSOR	
SCALE: FULL SIZE	DRAWING NO.
DATE: MAY 1, 1955	RTM- D3
ALL DIMENSIONS IN INCHES	

FIG. 6

PRESSURE TAP REFENCE NUMBER

1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23

$\frac{P}{P_{o_i}}$

1.0 0.9 0.8 0.7 0.6 0.5 0.4 0.3 0.2 0.1 0

RUN #26
RUN #27
RUN #28
RUN #29
RUN #30
RUN #31
RUN #32
#33

NOZZLE THROAT
WATER INJECTOR

TEST SECTION
DIFFUSER

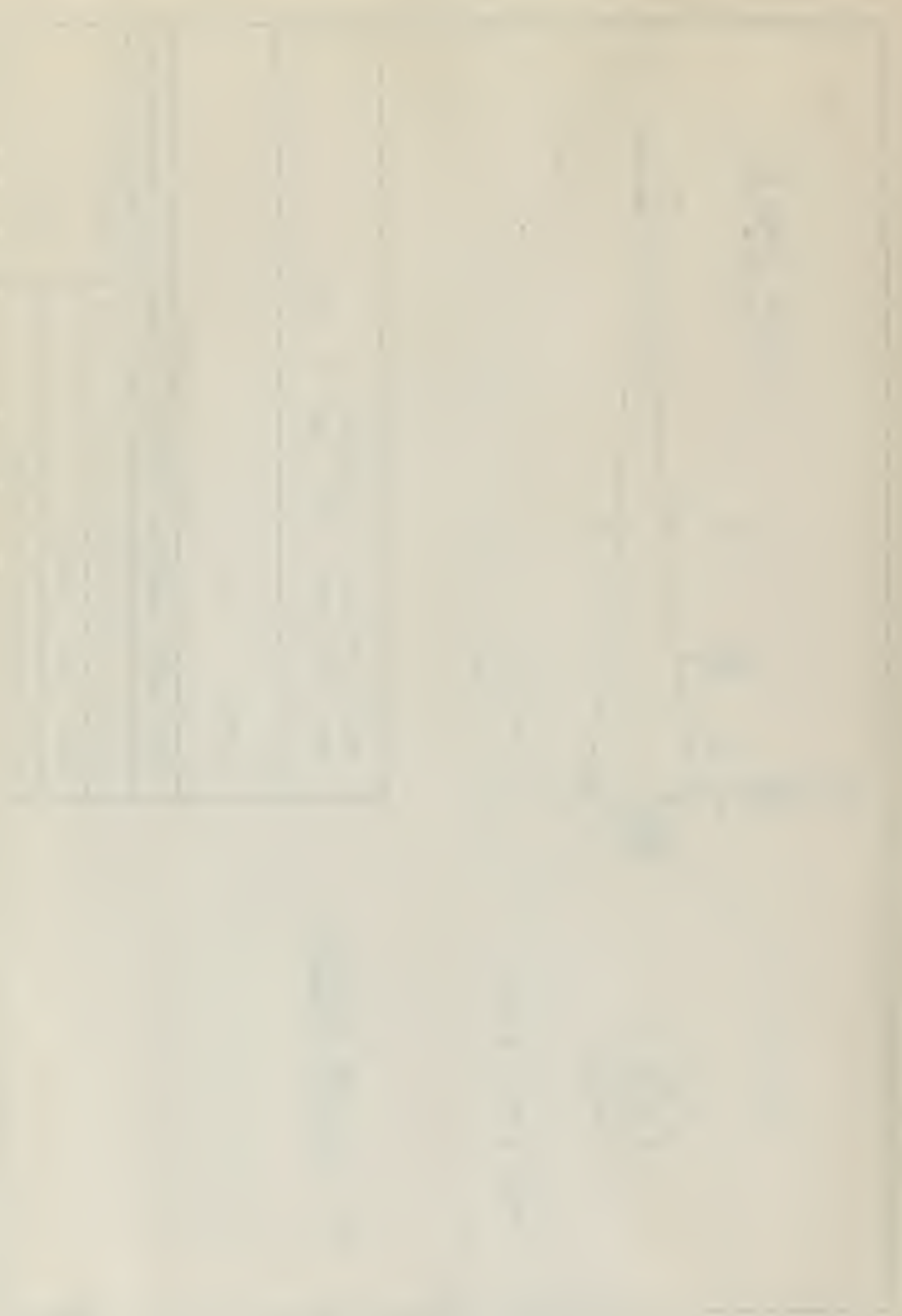
-15 -10 -5 0 5 10 15 20 25 30 35 40 45 51 55 60 65 70 75 80 85 90 95 100 105 110 115 120 125

DISTANCE DOWNSTREAM FROM BELLMOUTH ENTRANCE (INCHES)

AEROTHERMOPRESSOR PERFORMANCE, INLET MACH 1.5

T_{o_i} = 1040° F = 1500° R
T_{o_c} = 55° F
p_{o_i} = 14.5 psia.
W_a = .565 lb/sec.
W_g = .133 lb/sec.
Ω_o = W_g/W_a = .235

FIG. 7



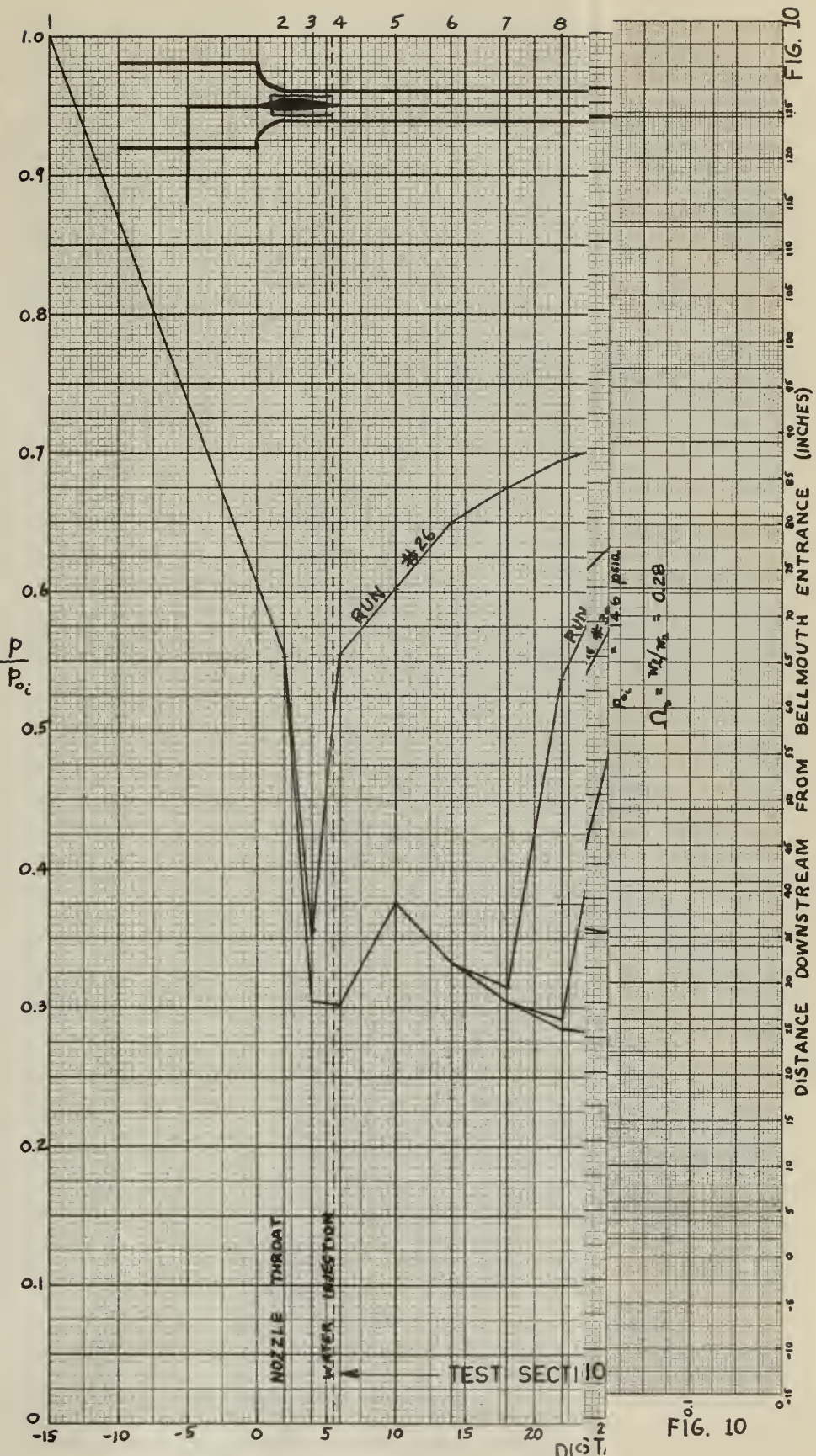

1
2
3
4
5
6
7
8
1.0
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
p/p0i
RUN #26
RUN #
p0i = 14.6 psia
Ωe = we/wa = 0.28
NOZZLE THROAT
WATER INJECTION
TEST SECTION
-15
-10
-5
0
5
10
15
20
DIST.
DISTANCE DOWNSTREAM FROM BELLMOUTH ENTRANCE (INCHES)

FIG. 10

FIG. 10

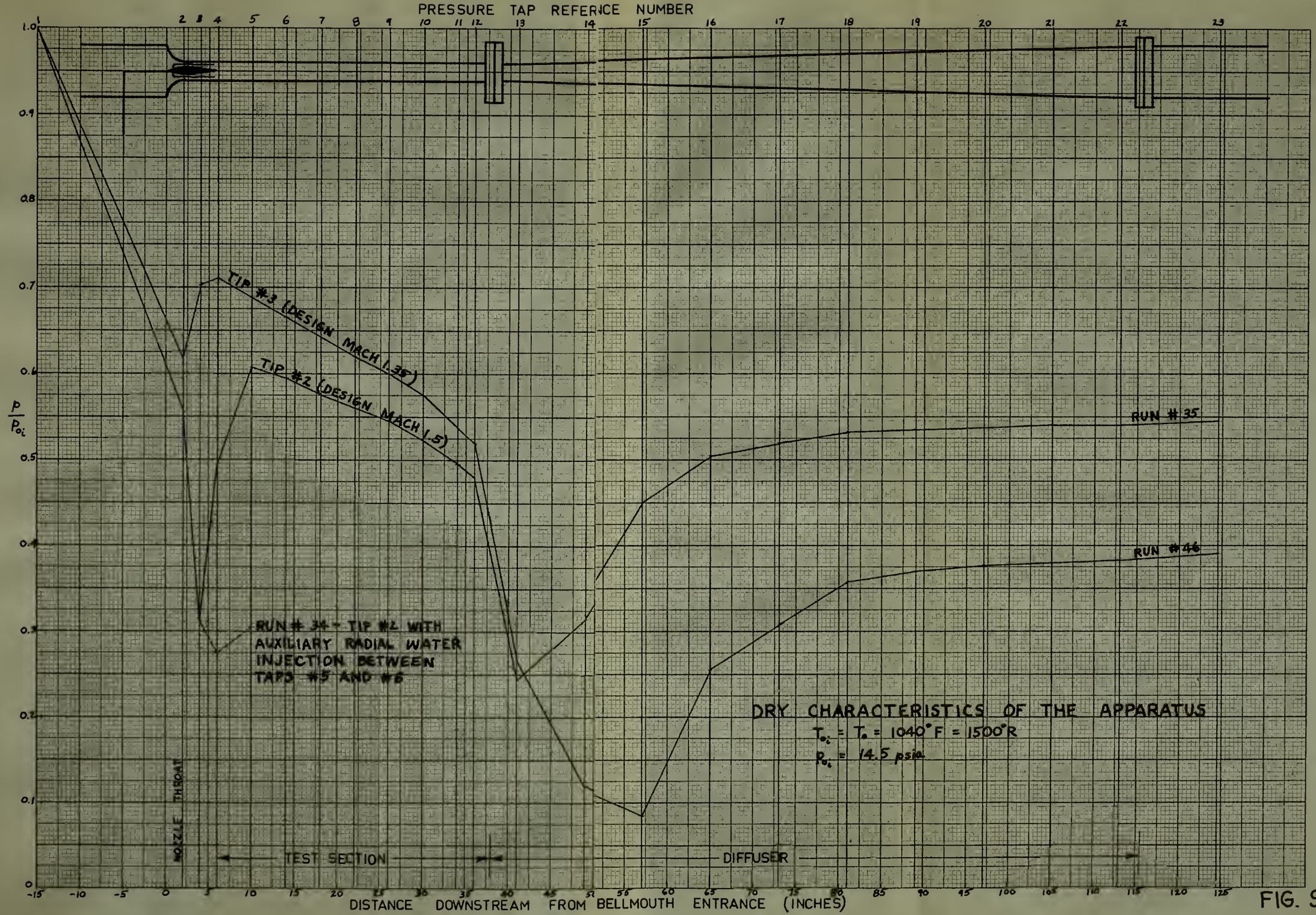

PRESSURE TAP REFERENCE NUMBER
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23
1.0
0.9
0.8
0.7
0.6
P/Po
0.5
0.4
0.3
0.2
0.1
0
TIP #3 (DESIGN MACH 1.35)
TIP #2 (DESIGN MACH 1.5)
RUN # 35
RUN # 46
RUN # 34 - TIP #2 WITH AUXILIARY RADIAL WATER INJECTION BETWEEN TAPS #5 AND #6
DRY CHARACTERISTICS OF THE APPARATUS
To = T∞ = 1040°F = 1500°R
Po = 14.5 psia
NOZZLE THROAT
TEST SECTION
DIFFUSER
-15 -10 -5 0 5 10 15 20 25 30 35 40 45 50 55 60 65 70 75 80 85 90 95 100 105 110 115 120 125
DISTANCE DOWNSTREAM FROM BELLMOUTH ENTRANCE (INCHES)

FIG. 9

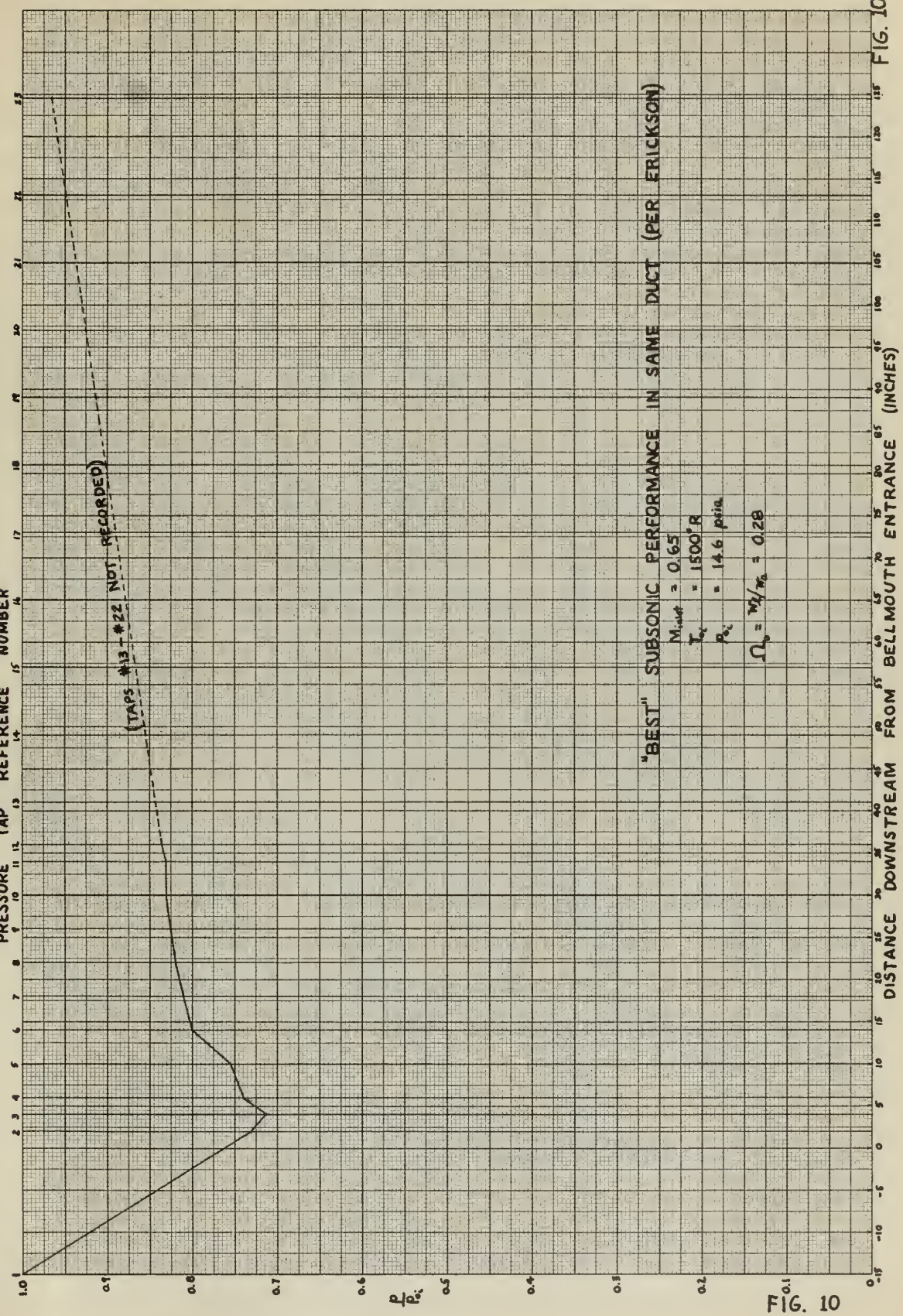

PRESSURE TAP REFERENCE NUMBER
DISTANCE DOWNSTREAM FROM BELLMOUTH ENTRANCE (INCHES)
p/p0
(TAPS #13 - #22 NOT RECORDED)
"BEST" SUBSONIC PERFORMANCE IN SAME DUCT (PER ERICKSON)
M throat = 0.65
T0 = 1500°R
P0 = 14.6 psia
Ω = W/W* = 0.28

FIG. 10

Thesis

28790

M213 MacKay

Experimental investigation of a 2 1/8"-diameter constant-area aerothermopressor with supersonic inlet.

Thesis

M213 MacKay 28790

Experimental investigation of a 2 1/8"-diameter constant-aera aerothermopressor with supersonic inlet.

thesM213
Experimental investigation of a 2 1/8 -
3 2768 001 88204 6
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